Performance Optimization of Storable Bipropellant Engines to Fully Exploit Advanced Material Technologies

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This paper summarizes the work performed to date on the NASA Cycle 3A Advanced Chemical Propulsion Technology Program. The primary goals of the program are to design, fabricate, and test high performance bipropellant engines using iridium/rhenium chamber technology to obtain 335 seconds specific impulse with nitrogen tetroxide/hydrazine propellants and 330 seconds specific impulse with nitrogen tetroxide/monomethylhydrazine propellants. Aerojet has successfully completed the Base Period of this program, wherein (1) mission and system studies have been performed to verify system performance benefits and to determine engine physical and operating parameters, (2) preliminary chamber and nozzle designs have been completed and a chamber supplier has been downselected, (3) high temperature, high pressure off-nominal hot fire testing of an existing state-of-the-art high performance bipropellant engine has been completed, and (4) thermal and performance data from the engine test have been correlated with new thermal models to enable design of the new engine injector and injector/chamber interface. In the next phase of the program, Aerojet will complete design, fabrication, and test of the nitrogen tetroxide/hydrazine engine to demonstrate 335 seconds specific impulse, and also investigate improved technologies for iridium/rhenium chamber fabrication. Achievement of the NRA goals will significantly benefit NASA interplanetary missions and other government and commercial opportunities by enabling reduced launch weight and/or increased payload. At the conclusion of the program, the objective is to have an engine ready for final design and qualification for a specific science mission or commercial application. The program also constitutes a stepping stone to future development, such as higher pressure pump-fed in-space storable engines.

NOMENCLATURE

ACS = *Attitude Control System*
AR = *Area Ratio*
*C** = *Characteristic Velocity of rocket combustion products*
CVD = *Chemical Vapor Deposition*
ΔV = *Delta Velocity*
FFC = *Fuel Film Cooling*
GEO = *Geosynchronous Earth Orbit*
GTO = *Geosynchronous Transfer Orbit*
IHPRPT = *Improved High Payoff Rocket Propulsion Technology*
Ir/Re = *Iridium lined Rhenium material system*
I_{sp} = *Specific Impulse*
JPL = *Jet Propulsion Laboratory*
kg = *Kilogram*
LAE = *Liquid Apogee Engine*
MMH = *Monomethylhydrazine*, $N_2H_3CH_3$
MON-X = *Mixed Oxides of Nitrogen, nitrogen tetroxide and X% NO by mass in solution*
MSFC = *Marshall Space Flight Center*
N_2H_4 = *Hydrazine*, N_2H_4
NRA = *NASA Research Announcement*
NTO = *Nitrogen Tetroxide*, N_2O_4
OF = *Oxidizer to Fuel ratio*
P_c = *Chamber pressure*
psia = *Pounds per square inch absolute*

I. INTRODUCTION

NASA's In-Space Propulsion Technology Program initiated the Cycle 3A Advanced Chemical Technology research announcement with the goal to increase the specific impulse of pressure-fed earth-storable bipropellant rocket engines to at least 330 seconds with NTO/MMH propellants and at least 335 seconds with NTO/N_2H_4 propellants. State of the art storable rocket engines deliver approximately 323 and 328 seconds I_{sp} for the respective propellant combinations given above[1,2]. Increased specific impulse has the obvious advantage of reducing the propellant required to perform spacecraft maneuvers. For telecommunications satellites in geosynchronous orbit the mass reduction can be applied to increasing the ACS propellant, hence life of the spacecraft, increasing the power generating capability and/or increasing the number of transponders, which increases the revenue potential for the satellite's owners and operators. For science missions, the propellant mass reduction can be applied to increasing the data gathering capability of the spacecraft. In some cases, where a direct cost benefit is not relevant, the capacity of the improved technology to enable a useful scientific mission which was not previously viable has been evaluated. Preliminary studies have indicated that a sufficient decrease in required propulsion system mass coupled with increased engine performance is likely to justify the further evolution of propulsion technology[3].

The goals of the Cycle 3A NRA program are aligned with Aerojet's in-space bipropellant engine technology development plan depicted in Fig. 1. Originally conceived to address IHPRPT goals for bipropellant engines, the plan is comprised of three spiral technology phases – (1) the full exploitation of pressure-fed storable engine technology to achieve 335 seconds Isp; (2) the development of high pressure systems, notionally via pumps, to achieve 345 seconds Isp; and (3) the development of engines with high energy oxidizers and/or additives to achieve 375 seconds Isp. Aerojet has recent and/or current programs addressing technology development in both spirals 1 and 2.

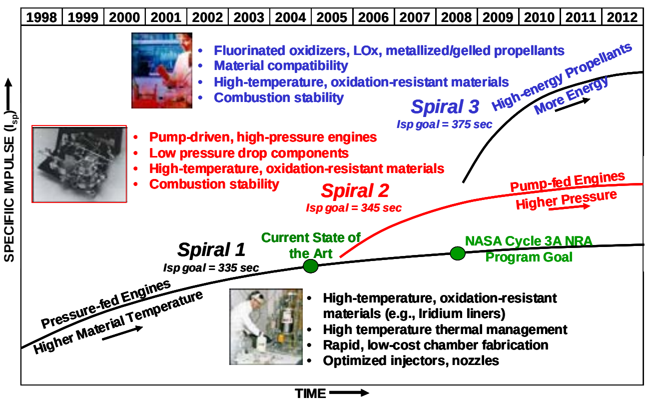

Fig 1. Aerojet's In-Space Bipropellant Engine Technology Development Plan

The NRA performance goals are expected to be achieved by expanding the operating envelope of flight-proven iridium/rhenium (Ir/Re) combustion chamber technology currently used for liquid apogee engines (LAEs), an example of which is shown in Fig. 2. This material system has the capacity to withstand steady-state wall temperatures approaching 2470ºK[4] compared to the state of the art usage at less than 1700ºK. The design approach is to modify Aerojet's state of the art HiPAT™ design such that the chamber wall materials operate closer to their temperature limits while maintaining safety margins critical to mission integrity. Changes to the current design will include injector optimization, chamber/nozzle contour optimization, reduced chamber emissivity, and increased thermal resistance between the injector and chamber. Engine operating conditions will also be modified (within mission constraints), to produce higher combustion gas temperatures. These will include higher feed pressure/lower internal pressure drop and higher/optimized mixture ratio. Fig. 3 graphically illustrates the design approach.

A secondary NRA goal is to investigate the viability of alternate iridium/rhenium fabrication processes and other related material systems to determine whether alternate processes offer cost, producibility, and/or performance advantages over the baseline chemical vapor deposition (CVD) Ir/Re fabrication process. If one of these alternate processes is deemed to be of sufficient value and level of development, it may be incorporated into one of the NRA program option engine designs.

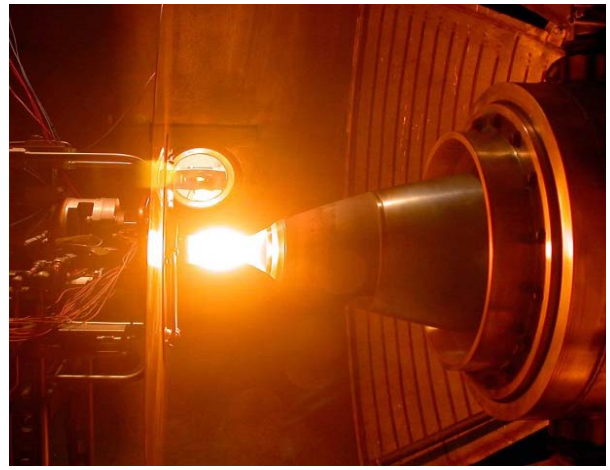

Fig 2. Iridium/Rhenium Liquid Apogee Engine Firing

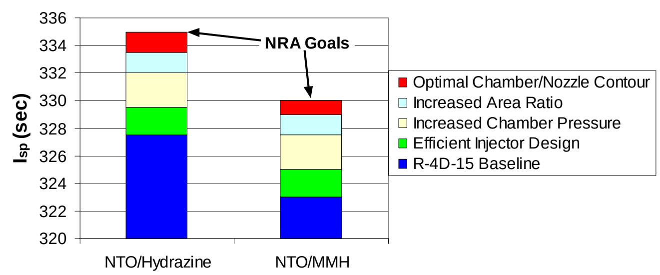

Fig 3. Design Approach to Achieve Cycle 3A NRA Program Goals

II. PROGRAM STRUCTURE

The overall program organization is shown in Fig. 4. Aerojet is responsible for program management, overall technical coordination, engine/component design and analysis, and engine testing and data reduction. JPL is a collaborator responsible for performing mission and spacecraft studies to assist in the development of engine performance and operating requirements. NASA MSFC is a collaborator responsible for propulsion system level studies to assist in the development of engine performance and operating requirements, and for the identification and evaluation of advanced chamber material technologies pursuant to the program's secondary goal. Technical Task Agreements were created to define and scope the JPL and MSFC work packages. Plasma Processes, Inc. (PPI) won a competitive downselect to provide the first combustion chamber for the new engine and will also be involved in the evaluation of alternative chamber technologies.

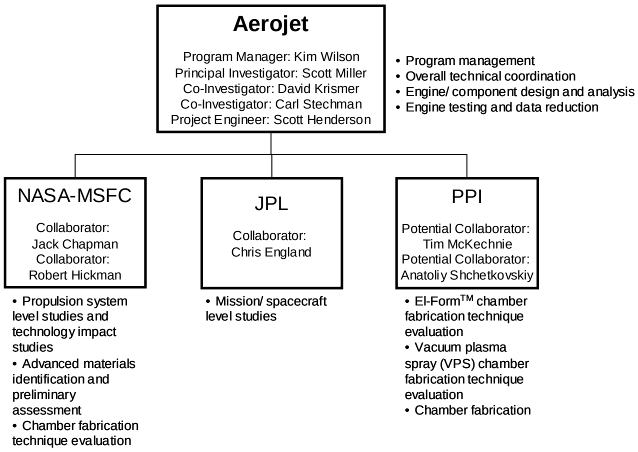

Fig 4. Cycle 3A NRA Program Organization

The NRA program is divided into three periods – the Base Period, Option 1 Period, and Option 2 Period. During the Base Period (September 2006-April 2007) which was recently completed, mission and system studies were performed to verify system performance benefits and to determine engine physical and operating parameters; preliminary chamber and nozzle designs were completed and a chamber supplier (PPI) was downselected for the Option 1 engine; high temperature, high pressure off-nominal hot fire testing of an existing state-of-the-art high performance bipropellant engine was completed; and thermal and performance data from the engine test were correlated with new thermal models to enable design of the new engine injector and injector/chamber interface. A program logic chart for Base Period activities is shown in Fig. 5.

During the Option 1 Period (April 2007-June 2008), engine design and analysis will be completed for both the NTO/N_2H_4 engine and the NTO/MMH engine; the NTO/N_2H_4 engine will be assembled and hot fire tested to verify program performance goals, and a materials evaluation study will be completed by MSFC to determine whether a different, more promising material technology will be used to fabricate the Option 2 engine combustion chamber. The program logic for the Option 1 Period is shown in Fig. 6.

In the Option 2 Period (June 2008-January 2009), the mission and system studies of JPL and MSFC are revisited to develop revised engine performance and operating requirements based on changes in NASA and commercial mission models and needs, and also on the results of the Option 1 engine testing. The Option 2 NTO/MMH engine is then assembled and tested, taking into account the revised performance and operating requirements. A program logic chart for the Option 2 Period activities is shown in Fig. 7.

At the conclusion of the program, the goal is to have both engine designs at TRL 6 or higher, ready for final design of interfaces and componentry based on specific customer needs, and ready to enter into a formal qualification program.

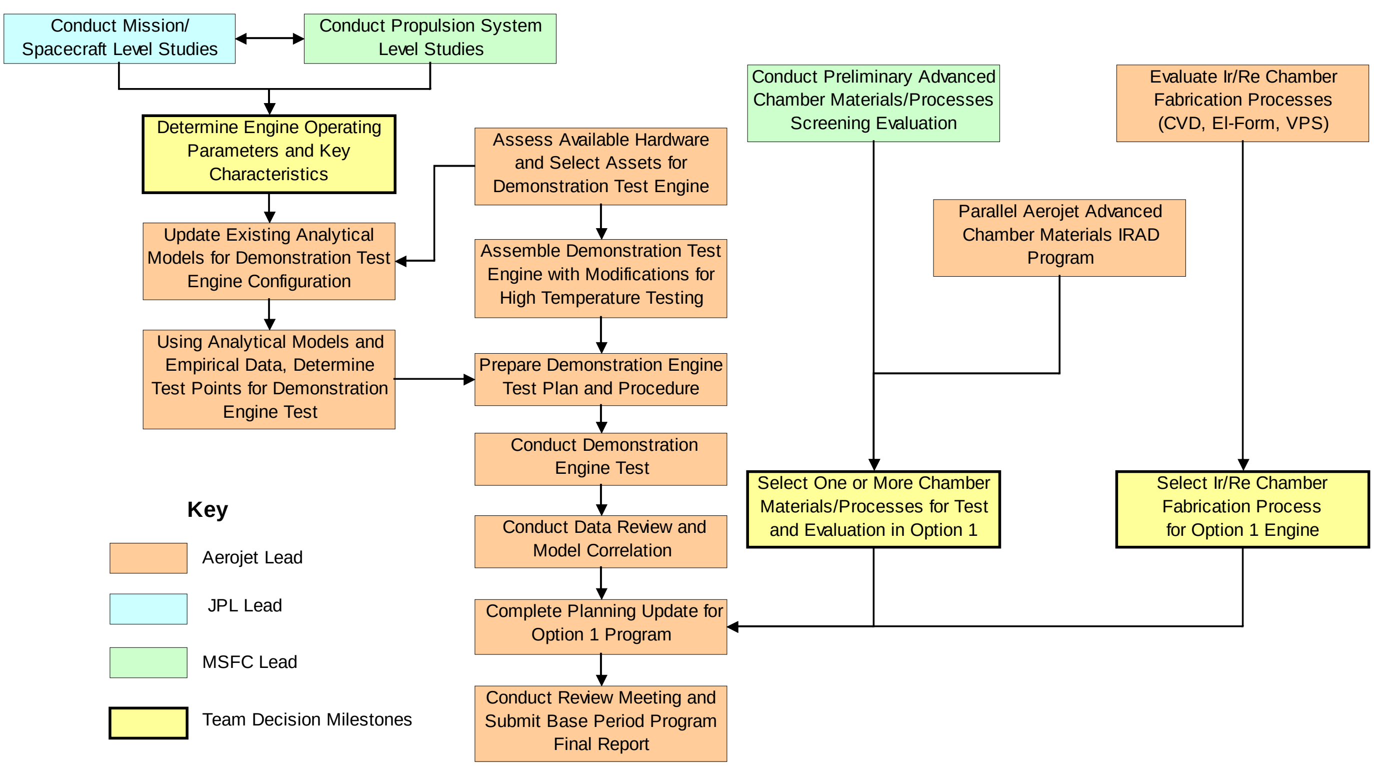

Fig 5. Base Period Program Logic Chart

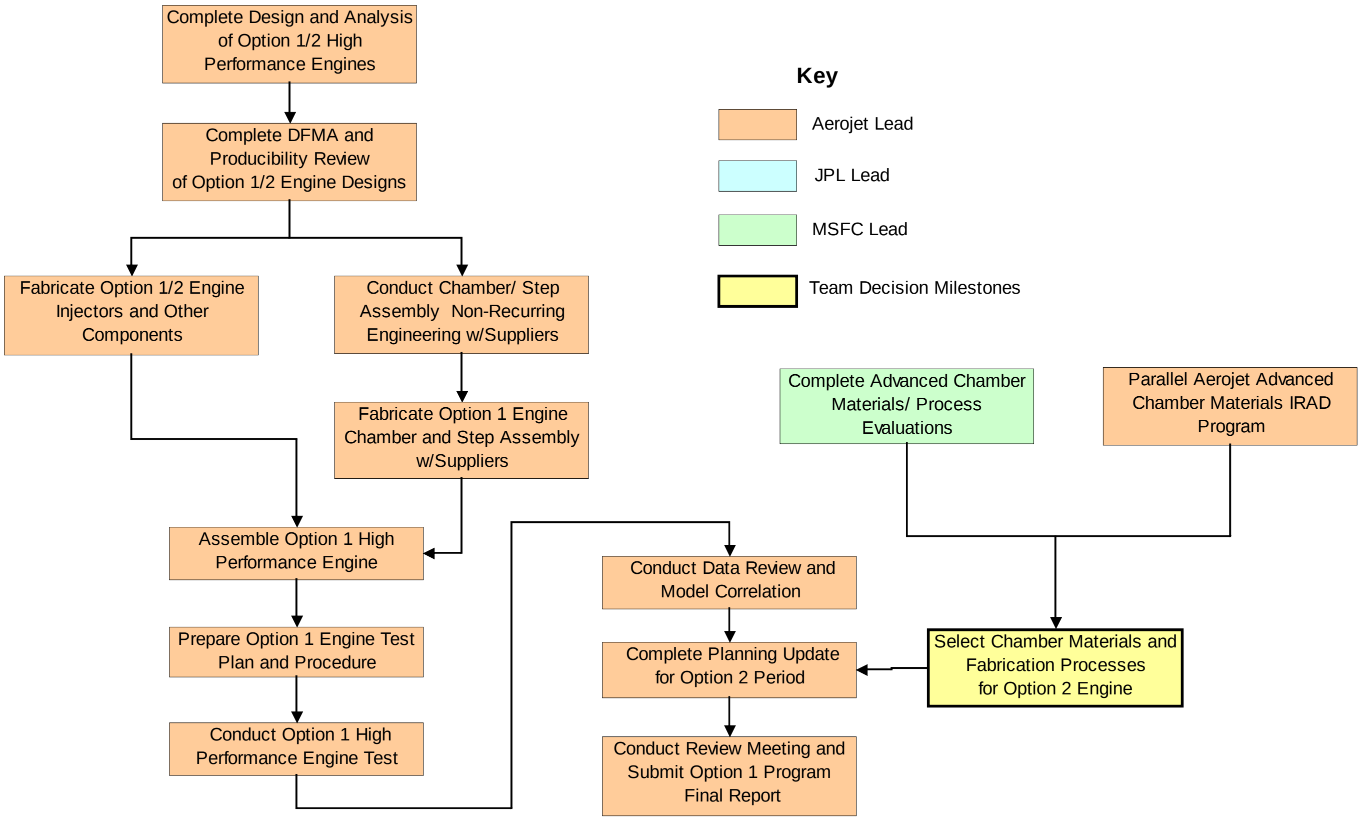

Fig 6. Option 1 Period Program Logic Chart

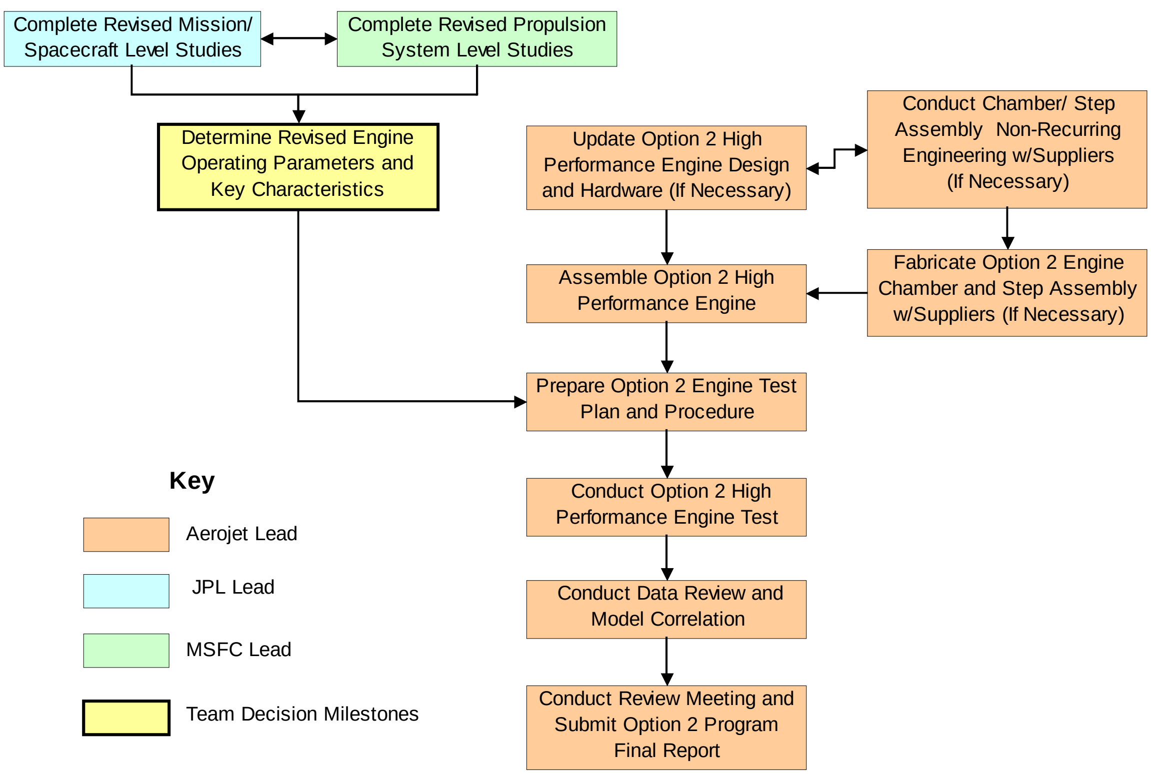

Fig 7. Option 2 Period Program Logic Chart

III. MISSION AND SYSTEMS ANALYSIS

For this program to be worthwhile, the benefits of achieving NRA goals must justify the cost. The benefits of improved system performance may be expressed in increased capability (payload) or improved finances (which usually flows directly from increased capability). In financial terms, there must be a great enough return on investment to amortize the research and development effort. For science missions, where an immediate financial benefit is not easily quantifiable, increased capability can enable a mission, allow use of a less expensive launch vehicle or increase a mission's scientific value.

To build confidence that there is a reasonable return on investment available, mission and system analyses were performed at NASA's Jet Propulsion Laboratory and at NASA's Marshall Space Flight Center. These analyses were intended to determine whether the integrated engine and propulsion system, presuming state of the art and near-term technology, is likely to permit a cost-effective increase in system capability.

A. Mission Analysis

The first step was a mission analysis for each of four reference missions, identified to be of commercial or current scientific interest, conducted primarily at JPL. The reference missions are identified in Table 1 with estimates of the launch mass, cumulative velocity change (ΔV) required of the Attitude Control System (ACS) and axial thrust elements of the propulsion system, and the mass of any deployed assemblies. The propulsion requirements for a GEO-sat were obtained from The Delft University of Technology[5], extrapolated for a 15 year service life. The ΔV required of a spacecraft's attitude control system can either be minimal, as is the case for many planetary missions, or have a significant effect on overall propulsion system size, as for the GEO satellite where station-keeping is a major system driver. Frequently, minimum system mass results from a dual-mode system, where the ACS uses hydrazine as a monopropellant and the axial engine burns hydrazine from the same supply system with NTO. Even if minimum mass does not result from a dual-mode system, economy of thrusters and improved reliability may mitigate in favor of such a system. For the purposes of comparison, dual mode axial/ACS propulsion was assumed for all spacecraft.

TABLE 1
REFERENCE MISSION SUMMARY DESCRIPTIONS

Mission	ACS need	Axial Δ-V, m/s	Launch Mass, kg	Deployed/ shed mass, kg
GTO to GEO	1,170 m/s	1,830	4,800	0
Europa Orbiter	23.4 kg	2,215	2,170	0
Mars Orbiter	20.0 kg	2,064	2,250	0
Titan-Enceladus (T-E) Orbiter	50.0 kg	2,368	6,633	1,298, 59.2 & 345

For each mission, the mass of the spacecraft at launch is estimated based on the expected launch vehicle capability and the terminal velocity which the launch vehicle is obligated to impart. The spacecraft trajectory is planned, in some cases taking advantage of planetary momentum exchange to modify the spacecraft velocity. Main engine burns are an essential part of trajectory planning to keep the spacecraft on course. In one case, the scientific requirements of the mission require deployment of spacecraft elements such as a heat shield or independent landing craft, requiring accounting for the mass decrements. Demands placed on the attitude control system are modeled based on historical data, acceptable limits of spacecraft pointing and statistical distributions of spacecraft attitude perturbations due to internal and external influences. The calculated propellant load is increased by 1% to account for the inability of propellant tanks to completely discharge their contents. Finally, because of the uncertainties inherent in engineering, a 5% margin is added to the propellant load.

Once the accounting is in place for mass and velocity changes, assumptions are made regarding the efficiency of the propulsion system elements. These assumptions are based on a database of past engine performance or in this case on the goals for improved main engine performance. The propellant mass required to execute the velocity changes required by trajectory planning and ACS analysis are determined by means of the rocket equation or similar calculation. Table 3 summarizes the propellant mass estimates calculated for the reference missions at the current state of the art I_{sp}, assumed to be 320 seconds for GEO missions and 325 seconds for planetary missions, and for main propulsion that achieves the NRA goals. The ACS I_{sp} was assumed to be 230 seconds for monopropellant hydrazine.

TABLE 2
PROPELLANT MASS ESTIMATES (KG) BY MAIN ENGINE ISP

Mission	Total Propellant load (kg) by Main Engine I_{sp} (in seconds)				
	320 sec	325 sec	330 sec	332.5 sec	335 sec
GTO to GEO	3,204	3,189	3,176	3,170	3,163
Europa Orbiter	N/A	1,131	1,120	1,116	1,109
Mars Orbiter	N/A	1,320	1,307	1,300	1,293
T-E Orbiter	N/A	2,969	2,942	2,928	2,914

B. Systems Analysis

High I_{sp} of an engine in isolation is insufficient to assert a system benefit because known means of increasing performance (e.g. higher chamber pressure, more energetic propellants, etc.) may require an increase in system mass or cost that negate the advantages of higher engine performance. Non-recurring costs to redesign and qualify an engine to realize higher performance must also be amortized over engine use unless public funds are available.

MSFC and JPL both maintain databases of propulsion system designs and have each developed a methodology for estimating the mass of components based on performance requirements. The model used by MSFC has been documented in some detail[6]. These models apply correlations for hardware mass based on propellant volume and storage pressure, thrust required and degree of redundancy to estimate the mass of components comprising the propulsion system. MSFC produced a representative system schematic for a single redundant system, from which the number and size of components for the representative missions were derived.

Propellant tank mass is almost always the largest element of system dry mass. While decreased propellant mass is expected to result in decreased mass of a tank to contain it, an increase in tank pressure required to feed a higher I_{sp} engine may lead to a net mass increase due both to thicker tank walls as well as more pressurant and a larger/stronger pressurant tank. For the increased performance options presented, it was assumed that the maximum required propellant tank pressure would be 400 psia with a 1.5 safety factor. Tank material was fixed as titanium (6Al-4V), ullage volume at 5% and a surface tension propellant management device (PMD) is assumed to add 10% to tank weight with 1% of the initial propellant load unusable. An assumption is implicit that the tank size may be closely optimized for the amount of propellant the spacecraft requires.

Pressurant tanks are often the next largest mass element of a propulsion system. Propellants are pressure fed from the tanks to the engine, so a composite-overwrapped helium pressure vessel was selected with size calculated assuming adiabatic blowdown of gas initially at 4500 psia down to a minimum regulator inlet limit of 800 psia.

For MSFC's system model, component masses are based on the mass of existing hardware that has been flight proven in the space environment (TRL 9[7]) in spacecraft like the Mercury Messenger or Space Shuttle. Additionally, 10% design contingency is applied to give confidence that system mass is not underestimated. Table 3 lists the system burnout masses estimated for each of the reference missions versus main propulsion specific impulse. Burn-out mass includes residual propellants as well as any unused ACS propellant at the time of main engine cut-off.

TABLE 3
PROPULSION SYSTEM BURNOUT MASS (KG) BY MAIN ENGINE ISP

Mission	System Burn-out Mass (kg) by Main Engine I_{sp}				
	320 sec	325 sec	330 sec	332.5 sec	335 sec
GTO to GEO	390.5	389.3	388.1	387.5	386.9
Europa Orbiter	N/A	161.0	160.2	159.9	159.4
Mars Orbiter	N/A	189.6	188.4	187.8	187.2
T-E Orbiter	N/A	331.5	329.1	327.9	326.7

Fig. 8 shows the predicted propellant and system burn-out masses for the four reference missions. As expected the Titan-Enceladus mission, with the highest ΔV need, has the steepest mass reduction with increased I_{sp}. The GEO mission has a relatively flat mass versus I_{sp} attributable not only to lower axial ΔV but to a much higher ACS requirement, which is also reflected in a proportionally higher burn-out mass compared to the similarly sized Titan-Enceladus flight. It may be noted that the Mars and Europa orbiters have significantly higher burn-out masses relative to propellant load than the larger spacecraft. It is typical for smaller systems to have higher inert mass fractions due to the fixed mass of components like pressure transducers, regulators and some valves which represent a proportionally smaller fraction of total system weight for larger systems.

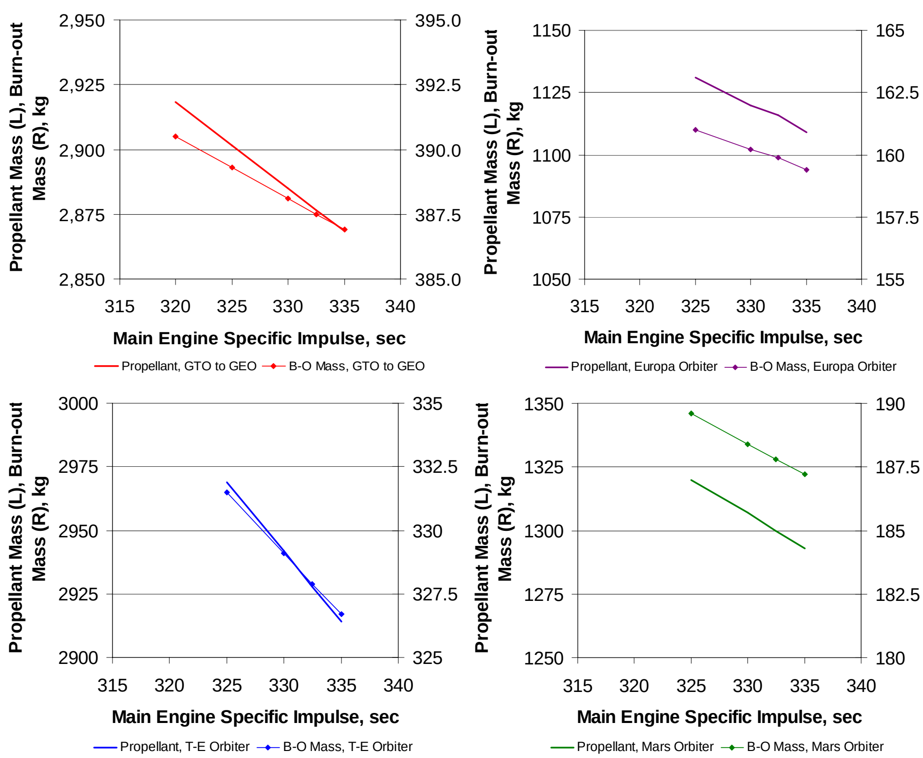

Fig. 8. Propellant and System Mass Reductions with Increased Specific Impulse

It can be concluded from the mission and systems analysis that significant reduction in spacecraft mass may be realized by improving engine specific impulse. A summary of total mass reduction is included in Table 4 compared to the baseline cases of 320 seconds Isp for the GEO mission and 325 seconds for the planetary missions. Compared to the total spacecraft mass these reductions are modest, however as a percentage of useful payload, they can be quite significant. For example, the Mercury Messenger spacecraft instrument payload is approximately 40 kilograms. For a 4800 kg GEOsat, 45 additional kilograms of propellant equates to an increase of useful revenue-earning life of approximately one year, based on the propellant usage and system masses backed out of the system model.

TABLE 4
TOTAL MASS REDUCTION SUMMARY

Mission	Total Propulsion System Mass Reduction				
	320 sec	325 sec	330 sec	332.5 sec	335 sec
GTO to GEO	0	16	30	37	45
Europa Orbiter	N/A	0	12	16	24
Mars Orbiter	N/A	0	14	22	29
T-E Orbiter	N/A	0	29	45	60

III. ENGINE OPERATIONAL AND PERFORMANCE REQUIREMENTS

In parallel with the Mission and Systems work performed by JPL and MSFC, Aerojet evaluated the design space for apogee class thrusters using internal design tools. The goal of this effort was to independently identify the trade space (i.e. chamber pressure vs. thrust level) that would yield engines having the targeted specific impulse of 335 seconds for hydrazine and 330 seconds for MMH.

Because the Option 1 engine (335 sec, NTO/hydrazine) will be the first one to be built, this was specified as the base configuration for both engines. The Option 2 engine (330 sec, NTO/MMH) will differ from the Option 1 engine only in the injector design (i.e. both engines share a common chamber and nozzle design). The results of the Option 1 engine analysis are shown in Fig. 9. The region bounded by the red box defines the range of “reasonable” answers. Engines outside this box become excessively large, in physical size, at lower chamber pressures and unrealistically small at the higher chamber pressures. The nominal design point of 200 lbf thrust, 275 psia chamber pressure at a mixture ratio (OF) of 1.2 was selected based on internal Aerojet GTO-to-GEO trade studies.

The Option 2 engine using this common chamber and nozzle has a predicted specific impulse of 331.6 sec at a mixture ratio of 1.94 (201 lbf thrust and 275 psia chamber pressure). Both designs use a 400:1 area ratio nozzle to maximize the performance benefit of increased area ratio while still remaining within the physical envelope of current HiPAT™ engines. The design point conditions are summarized below in Table 5. The results of this effort were then compared to the Mission and Systems Analysis results and were found to agree favorably.

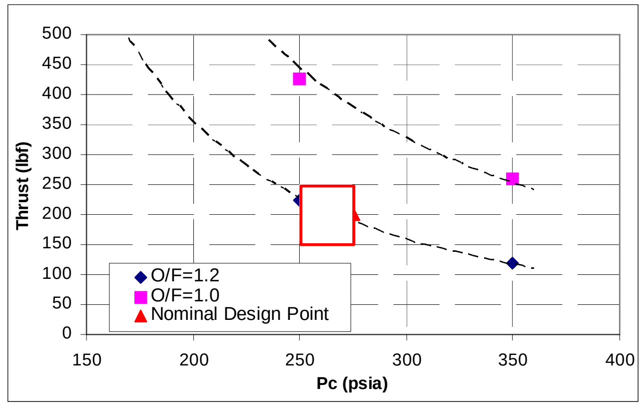

Fig. 9. Thrust vs. Chamber Pressure Design Space for the Option 1 Engine, Design Point Specific Impulse of 335 Seconds

TABLE 5
OPTION 1 & 2 ENGINE DESIGN POINT OPERATING CONDITIONS

	Isp, seconds	F, lbf	Pc, psia	MR, O/F	Pfeed, Psia	Nozzle Area Ratio
Option 1 (N2H4/NTO)	335	200	275	1.2	400	400:1
Option 2 (MMH/NTO)	331.6	201	275	1.94	400	400:1

IV. CHAMBER MATERIALS/PROCESS SELECTION

This task consisted of two distinct parts. The first task was performed by Aerojet to conduct an assessment of existing Ir/Re fabrication methods and to downselect a specific process and supplier for the Option 1 engine combustion chamber. The second task was performed by MSFC and focused on the selection of candidate materials to be evaluated during the Option 1 phase of the program for potential incorporation into the Option 2 engine combustion chamber.

A. Chamber Materials Selection

In order to conduct the chamber materials selection for the Option 1 engine combustion chamber, a preliminary engine design had to be completed so that suppliers could appropriately evaluate process capabilities against the expected chamber diameter, wall thickness, length and contouring. Based on the derived engine operational and performance requirements, Aerojet completed a preliminary design of the chamber and nozzle consistent with the performance goals of the program. The resulting overall external engine dimensions are shown in Fig. 10.

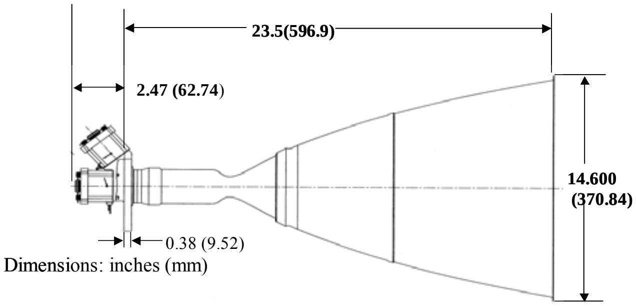

Fig. 10. Preliminary Option 1 & 2 Engine Envelope

The Option 1 chamber process selection involved the assessment of Ir/Re thrust chamber fabrication processes. These processes included Chemical Vapor Deposition (CVD), electroforming (El-Form), Low Pressure Plasma Spray (LPPS) and Vacuum Plasma Spray (VPS). Aerojet evaluated these processes, visited fabrication facilities, and conducted a competitive analysis of the methods for fabrication of typical in-space rocket engine thrust chambers.

Of the processes listed above, CVD is the incumbent process that Aerojet Redmond uses to fabricate the R-4D-15 HiPAT™ thrust chambers and is very well understood at this point. The only other process that Aerojet Redmond has used to fabricate an Ir/Re chamber for a bipropellant engine is El-Form. Neither LPPS nor VPS have been used by Aerojet for this application before. LPPS and VPS were therefore dropped from consideration for the Option 1 chamber due to the lack of technical maturity.

The Figures of Merit that were used for the decision matrix were:

- Cost – Nonrecurring
- Cost – Recurring
- Schedule – Nonrecurring
- Schedule – Recurring
- Producibility
- Performance – Mechanical Properties
- Performance – Thermal
- Performance - Oxidation Resistance
- Performance - Mass
- Heritage/Risk – Design
- Heritage/Risk - Manufacturing

Weighting factors were assigned to the Figures of Merit based on the primary performance goals of the program.

The PPI El-Form process was downselected due primarily to the lower development unit costs and production cost estimates. The El-Form process does carry more process risk since it is not as well developed as CVD, however the added risk is deemed worth the potential rewards in reduced costs.

B. Material/Process Evaluation Selection

The purpose of this task was to consult on the available materials and fabrication processes for use in high temperature rocket engine thrust chambers to determine whether there were any candidates for further evaluation during the Option 1 Period for potential incorporation in the Option 2 engine combustion chamber. The scope of the effort was limited to previously existing materials, processes, and data sources. Table 6 summarizes the candidate materials selected and their pros and cons[8]. Upon completion of the study, MSFC made a recommendation to Aerojet on a materials/manufacturing process to be evaluated during the Option 1 program (as sample coupons). At this time the recommendation is still under evaluation.

TABLE 6

OPTION 1 MATERIALS TESTING CANDIDATE LIST

Candidate Material/Process	Cost/Schedule	Pros	Cons
Engineered EL-Form Rhenium (Re)	▪ High (but likely in scope)	▪ Improved YS and UTS	▪ Process repeatability compared to traditional EL-Form Re not known ▪ Elevated temperature strength and life not known
Thick EL-Form Ir layer	▪ Moderate	▪ Improved life	▪ Increased chamber life not demonstrated
Functionally graded ceramic lined Ir/Re (HfO_2 or ZrO_2)	▪ Out of Scope	▪ Increased operating temperature and life ▪ Improved oxidation resistance	▪ Low TRL ▪ Significant process development/testing needed
VPS Ir/Re	▪ Out of Scope	▪ Low cost ▪ High tensile strength ▪ Thicker Ir	▪ Significant process development and testing required
HIP bonded Ir /Re	▪ Out of Scope	▪ Thicker Ir ▪ Improved Re process repeatability ▪ Improved properties	▪ Thrust chamber fabrication process not well known ▪ Complex fabrication ▪ High cost
Dispersion strengthened Mo/Re or other advanced alloy	▪ Out of Scope	▪ Reduced weight ▪ Lower cost ▪ Improved elevated temperature strength	▪ Significant development and testing required ▪ Joining characteristics not well known

V. BASE PERIOD ENGINE TEST

To increase the knowledge base prior to designing and testing a new, higher-performance engine, an experimental investigation was conducted to map the performance of an existing Aerojet R-4D-15 Ir/Re engine over a range of chamber pressures and mixture ratios. This engine is optimized to operate best at feed pressures of 219 to 310 psia and NTO/N_2H_4 OF ratios of 0.716 to 1.188[3], which are both lower than expected for a next generation engine. No effort was made to optimize the test unit for the new conditions. While performance can be expected to increase due to operation at higher P_c and OF, the results of this test can not be expected to be as good as if an engine were optimized to operate at these conditions.

The test engine, shown in Fig. 11, was derived from existing Aerojet engine assets, including workhorse propellant valves. The engine was instrumented within the walls of an internal step assembly[2] with thermocouples to measure the effectiveness of FFC during firing. The intent was to learn how far OF can be increased before FFC is no longer effective. When the FFC was effective, step temperatures were at approximately the saturation temperature of N_2H_4 at P_c. This indicates that there is normally a two-phase fuel film on the step. Knowledge of the conditions under which FFC breaks down will be used to iterate the injector design and optimize C* while still providing adequate thermal protection for the engine.

Fig. 11. Base Period Test Engine Instrumented with Thermocouples

The test matrix was designed to seek the maximum attainable chamber pressure for three mixture ratios (MR=0.85, 1.0 & 1.1) as shown in **Error! Reference source not found.** 12. For each mixture ratio the engine would start at a nominal 100 lbf thrust condition. Once thermal equilibrium was achieved (approximately 60 seconds), the propellant feed pressures were increased in set increments followed by a dwell time of 30 seconds minimum at each pressure to allow the engine to reach quasi-thermal equilibrium. Steps in pressure were defined based on the amount of propellant allocated to the test. This process continued until one of the pre-determined limits was reached.

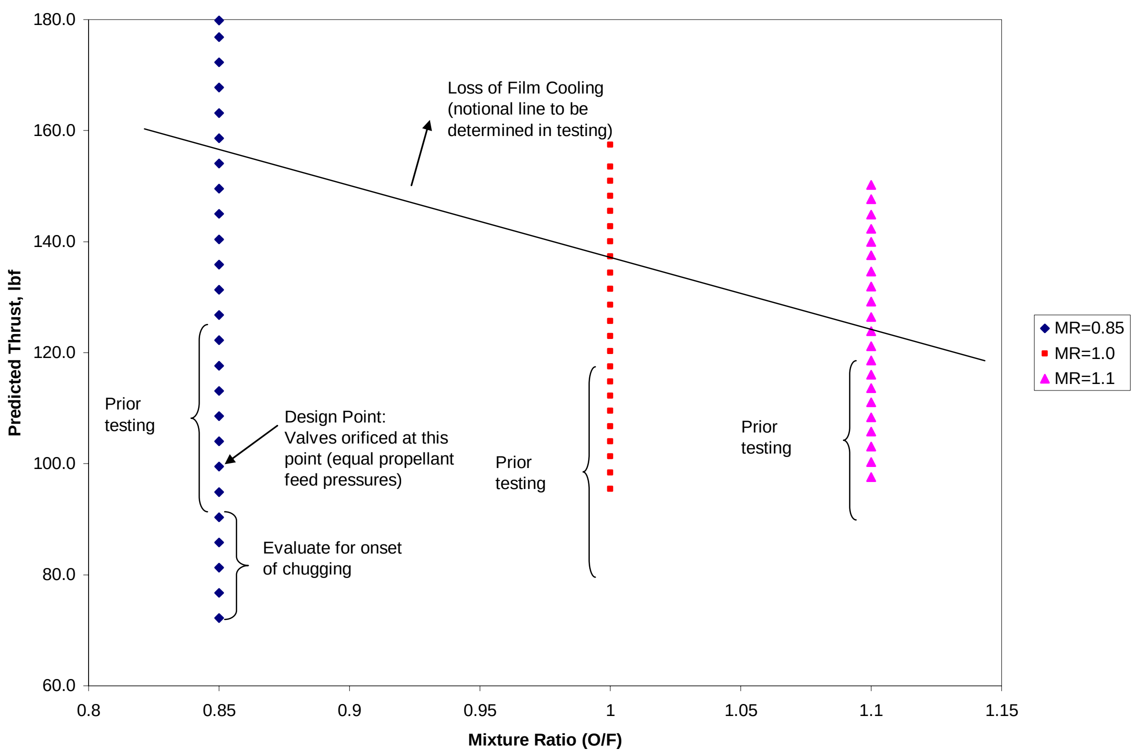

Fig. 12. Base Period Engine Hot Fire Test Matrix

Test limits were defined by:

- Maximum facility propellant supply pressure
  - ~470 psia for the fuel
- Thermal limits of the engine
  - Thermocouples TC1 through TC4 and Chamber throat pyrometer monitored during testing with automated shutdown at preset limits (750ºF for TC1-4 and 4000ºF for the pyrometer)
  - Loss of fuel film cooling of the step (manual cut based on real-time step temperature monitoring or automated cut based on soak back to TC1-4)
- Combustion roughness (manual shutdown by test engineer)

The test program was very successful, as all goals were achieved and the data was of high quality. The primary test highlights are summarized below:

- 26 hot fire runs over four days
- 2909 seconds of total burn time
- Propellant consumption
  - 472 lbm NTO
  - 569 lbm N2H4
  - 156 lbm N2H4
- 3673ºF (2296ºK) maximum chamber temperature
- 217 psia maximum chamber pressure (prior maximum of 160 psia)
- 53.4 psia minimum chamber pressure (prior minimum of 99.4 psia)
- 329 seconds maximum specific impulse (prior maximum of 328.3 sec)
- Platinum step temperatures successfully collected (data not previously obtained)
- 2D exterior chamber/nozzle temperature distribution successfully collected via new IR camera (data not previously obtained)

A summary of the wide range of test data gathered as a function of mixture ratio and thrust for all tests conducted by Aerojet is shown in Fig. 13. An example of the 2D IR camera data obtained is shown in Fig. 14.

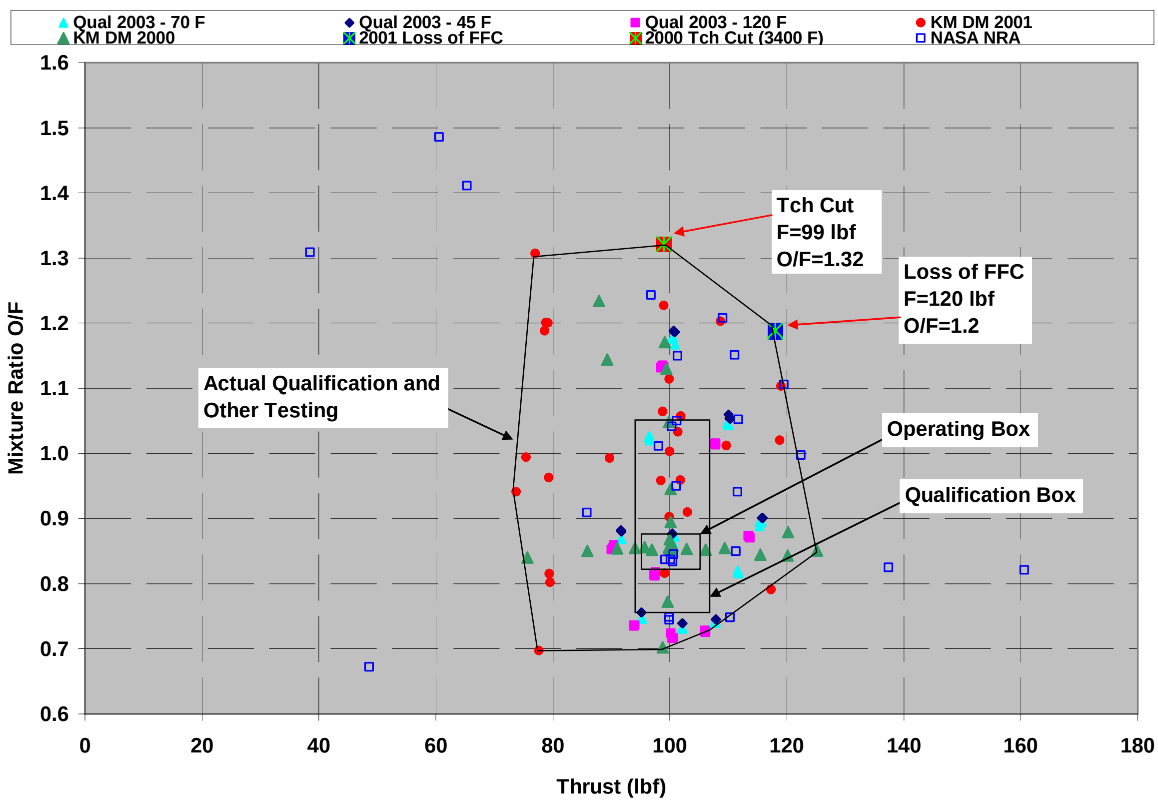

Fig. 13. R-4D-15 NTO/N2H4 Test Data Map

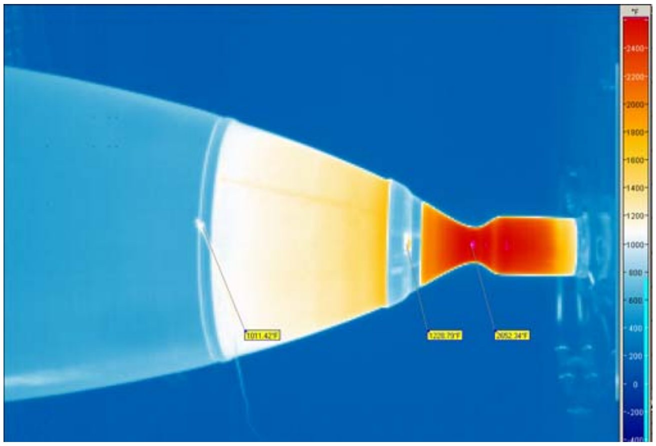

Fig. 14. IR Camera Image of Base Period Engine Test Firing

Another goal of testing was to determine the minimum delta pressure across the injector required to prevent chug, defined here as low frequency chamber pressure oscillation coupled with feed system pressure. As chamber pressure is increased to improve performance, a correspondingly large feed pressure increase is undesirable for its weight increase on system components. Demonstration of chug delayed to very low feed pressure provides evidence that injector delta pressure may be decreased safely. The test engine was successfully tested by stepping down chamber pressure until sustained chugging was observed at approximately 80 psia at a frequency of approximately 178 Hz. This is much lower than nominal operating P_c indicating that the inlet pressure range of 219 to 310 psia[2] might be reduced, or P_c increased, by lower injector pressure drop without risk of related instability.

VI. CONCLUSIONS

Aerojet and its collaborators have successfully completed the Base Period phase of the NASA Cycle 3A Advanced Chemical Propulsion Technology Program. The primary goals of the program are to design, fabricate, and test high performance bipropellant engines using iridium/rhenium chamber technology to obtain 335 seconds Isp with NTO/N2H4 propellants and 330 seconds Isp with nitrogen NTO/MMH propellants. Mission and system studies have been performed to verify system performance benefits and to determine engine physical and operating parameters, preliminary chamber and nozzle designs have been completed and a chamber supplier has been downselected, high temperature, high pressure off-nominal hot fire testing of an existing state-of-the-art high performance bipropellant engine has been completed, and thermal and performance data from the engine test have been correlated with new thermal models to enable design of the new engine injector and injector/chamber interface. In the next phase of the program, Aerojet will complete design, fabrication, and test of the NTO/N_2H_4 engine, and also investigate improved technologies for iridium/rhenium chamber fabrication. Achievement of the NRA goals will significantly benefit NASA interplanetary missions and other government and commercial opportunities by enabling reduced launch weight and/or increased payload. At the conclusion of the program, the objective is to have an engine ready for final design and qualification for a specific science mission or commercial application.

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